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THE CHARACTERISTICS OF PLAIN FLAPS

This, Probably, is the First Time the Question of Tailplane Characteristics with Floating Elevators Has Been Dealt With. This Condition is Important in Tail-setting and Stability Calculations

By W. R. ANDREWS, A.F.R.Ae.S.

It has been recognised for many years that the theoretical characteristics of plain flaps are not obtained on full-scale tests. Mostly these tests are made on tailplanes fitted with elevators, and to make any direct comparison with the theoretical values one must have an accurate knowledge of the direction of the airflow at the tailplane as well as its velocity. Even when the airflow conditions are known, the measurement of elevator angles is not easy, unless special precautions are taken to allow for the deformation of parts under load. It is suspected that, in certain tests, this deformation has been included in the "efficiency" of the tailplane. Results from one series of tests can then only be applied to the design of a similar tailplane.

The use of even a small amount of elevator introduces considerable twisting on the tailplane, which must in most designs appreciably affect the incidence. The direct comparison between theory and practice is masked by such secondary effects. Even allowing liberally for these, the full-scale results still fall below the calculated values.

From consideration of the tailplane itself, the cause of the loss in efficiency can be divided into three headings :—

(1) Interference upon the spanwise distribution of lift across the tailplane due to :—

(*a*) discontinuous elevators.

(*b*) interference of the body.

(2) Errors in the theoretical treatment due to :—

(*a*) discontinuity at elevator hinge,

(*b*) the effect of tailplane thickness/chord ratio.

(3) The gap between the elevator and fixed portion which is not allowed for in the theoretical treatment.

It is obviously impossible to generalise on the effects under heading (1), but some attempt will be made to reduce the discrepancies, and to show the mechanism of the flow round the tail plane due to the gap at the hinge.

In a previous article (Ref. 1) empirical corrections to the theoretical characteristics of thin aerofoils (Ref. 2) are evolved, the results being summarised below.

THE NO-LIFT MOMENT.

$$K_{m_0} = f(t)\left[-\frac{\pi}{4}\epsilon' + \mu_0 + .0036\phi - .0453\psi)\right] \quad .. \quad (1)$$

Where $f(t) = 1 - 4.9\,t^2$.

t = Maximum thickness – chord ratio

$\epsilon' = \int_0^1 y_0 f_1(x)\,dx$

y_0 = "Rise" of centreline of aerofoil with respect to LE – TE line and measured normal to centreline itself. (Expressed as fraction of chord length.)

$$f_1(x) = \int_0^1 \frac{1}{\pi(1-x)\sqrt{x(1-x)}}$$

ϕ = Slope of centreline at LE.

ψ = Slope of centreline at TE.

ϕ is usually positive and ψ is negative except for aerofoils having reflexed trailing edges.

NO-LIFT ANGLE.

$$\alpha_0 = \frac{57{\cdot}3}{\pi/4}\left[K_{m_0} - \frac{\mu_0}{f(t)}\right] - 10\gamma \quad .. \quad .. \quad (2)$$

$\mu_0 = \int_0^1 y_0 f_2(x)\,dx$

$$f_2(x) = \frac{1 - 2x}{\sqrt{x(1-x)}}$$

γ = Maximum "rise" of centreline expressed as a fraction of the chord.

Slope of Lift Curve. (Infinite Aspect Ratio).

$$a_0 = \frac{dK_L}{d\alpha} = .0506 - .565\,t^2 \quad .. \quad .. \quad .. \quad (3)$$

Slope of Moment Curve.

$$e = \frac{dK_m}{dK_L} = .25 - b\, t^2 \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (4)$$

Where b is given in Fig. (1).

The theory of thin aerofoils has been extended to cover the aerofoil fitted with flaps (References 2 and 3). The notation employed is shown in Fig. 2 and Table 1.

Then

$$y_0 = \frac{hx}{1 - E} \text{ from } x = 0 \text{ to } (1 - E)$$

$$= \frac{h(1 - x)}{E} \text{ from } (1 - E) \text{ to } 1.0.$$

These relationships apply only where the flap angles are small.

In order to show how nearly the approximate relationships agree with the actual, Figs. 3 to 7 have been prepared. The dashed lines in each case refer to the approximate relationships. Except for E′ at small values, the approximate values are correct to within 1 per cent. for flap angles up to 15°. For small flaps, the ratio of effective flap chord to actual may cause a serious error. Only wind tunnel tests or a refinement to the theoretical treatment will confirm whether the actual (E) or the effective (E′) should be used for calculating the characteristics. It is necessary that any wind tunnel data used for this purpose should be carried out at high Reynolds Number.

TABLE I.

	APPROXIMATE EXPRESSION.	ACCURATE EXPRESSION.
C′	C	$(1 - E)\cos\alpha_1 + E\cos(\eta - \alpha_1)$
E′	E	$\dfrac{E\cos(\eta - \alpha_1)}{C'}$
α_1	$E\eta$	$\operatorname{Tan}^{-1}\dfrac{E\sin\eta}{1 - E(1 - \cos\eta)}$
γ	$E(1 - E)\eta$	$\dfrac{(1 - E)\sin\alpha_1}{(1 - E)\cos\alpha_1 + E\cos(\eta - \alpha_1)}$

The theoretical values for the plain flapped aerofoil (empirically corrected in line with equations 1 and 2 for the unflapped aerofoil) are as follows :—

No-lift Moment.

$$\triangle Km_0 = f(t)[-m\eta + .0036\phi - .0453\psi]$$
$$= f(t)[-m\eta + .0036\,E\eta + .0453\eta(1 - E)]$$

where $m = (1 - E)\sqrt{E(1 - E)}$

η = flap angle in radians

or with η expressed in degrees we have

$$\triangle Km_0 = \frac{\eta^\circ}{57.3} f(t) f_1(E) \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (5)$$

where $f_1(E) = [-(1 - E)\sqrt{E(1 - E)} + .0036\,E - .0453\,(1 - E)]$

No-lift Angle.

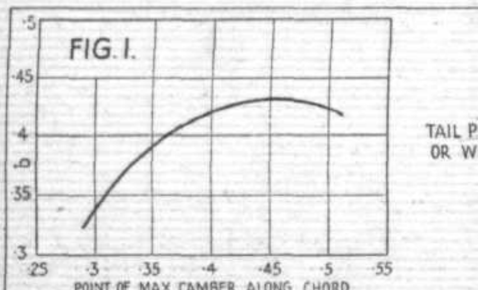
$$\triangle\alpha_0 = 57.3\left[\triangle Km_0 - \frac{\mu_0}{f(t)}\right] - 10\gamma - \alpha \text{ degrees}$$

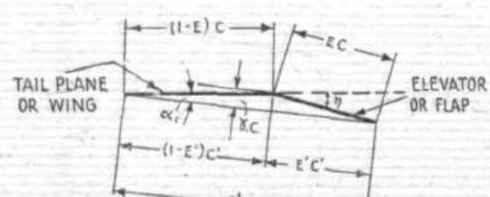
$$= 57.3\left[\triangle Km_0 - \frac{\mu_0}{f(t)}\right] - \frac{\eta}{5.73} E(1 - E) - E\eta$$

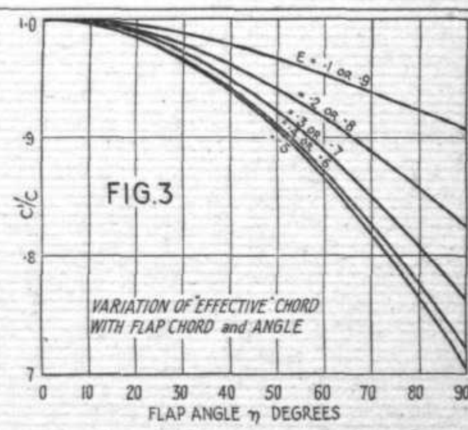
where η is now in degrees.

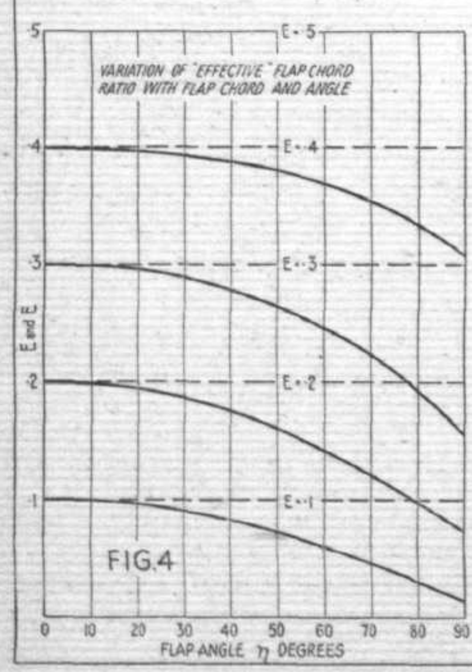
$$= \eta\left[f(t)f_1(E) - \frac{\mu'_0}{f(t)} - \frac{E}{5.73}(1 - E) - E\right] \ldots \quad (6)$$

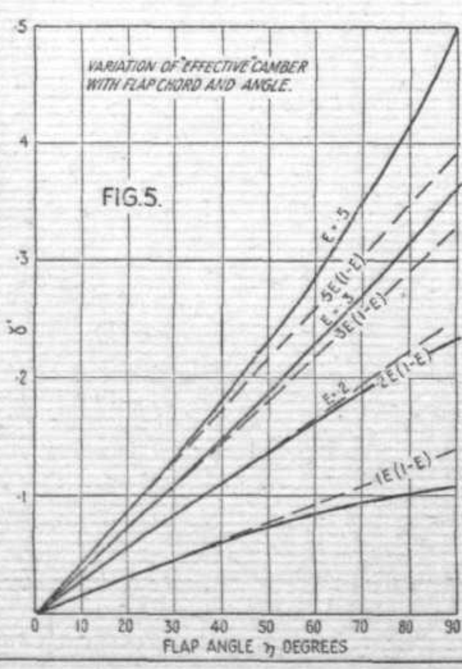
where $\mu'_0 = \dfrac{E(1 - E)}{2}\left[\dfrac{2E - 1}{\sqrt{E(1 - E)}} + \dfrac{\pi}{2E} - \dfrac{\operatorname{Arc\,Cos}\sqrt{E}}{E(1 - E)}\right]$

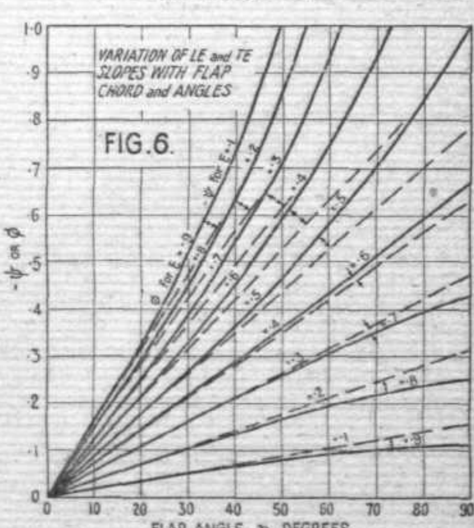












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The values of f_1 (E) and μ'_0 are plotted in Figs. 8, 9, 10 and 11.

It will be noticed that equation 5 has a small error at both ends and does not quite satisfy the conditions for $E = 0$ and $E = 1.0$. This is shown clearly on Fig. 9. Since it is unlikely that a flap (even a trimming tab) will be smaller than .03 of the main chord no attempt has been made to correct this very small inaccuracy in the equation.

The lift of the tailplane can be expressed as

$$K_L = a_1 \alpha + a_2 \eta \quad \ldots \quad (7)$$

where a_1 = slope of lift curve with elevators fixed.
α = Incidence of fixed portion.
a_2 = Slope of lift curve due to change of elevator angle (η) only.

If it is assumed that for flap angles up to 15° the slope of the lift curve (a_1) remains constant then the ratio

a_2/a_1 is given by $-\dfrac{\triangle a_0}{\eta}$

This ratio is plotted in Figs. 12 and 13 for 10 per cent. thickness to chord ratio. This curve suffers the same disability as Figs. 8 and 9 at $E = 0$ and $E = 1.0$.

The total moment about the LE of the aerofoil is then given by

$$Km = Km_0 - eK_L \quad \ldots \quad (8)$$

Where e is given by 4.

The only other characteristic is the hinge moment coefficient which is expressed in the form

$$K_H = \frac{e}{.25} \cdot \frac{b_1}{a_1} K_L - b\frac{\eta}{57.3} + \frac{.0453 \cdot \eta \cdot (1 - E)}{57.3} \quad \ldots \quad (9)$$

Where

$$-\frac{b_1}{a_1} = \frac{1}{\eta E^2}\left\{\left(\frac{3}{2} - E\right)\sqrt{E(1 - E)} - \left(\frac{3}{2} - 2E\right)\left(\frac{\pi}{2} - \text{ArcCos}\sqrt{E}\right)\right\}$$

and
$$b = \frac{2(1 - E)\sqrt{E(1 - E)}}{\pi E^2}\left\{\frac{\pi}{2} - \text{ArcCos}\sqrt{E} - \sqrt{E(1 - E)}\right\}$$

Equation 9 can be rewritten as

$$K_H = \frac{e}{.25}\frac{b_1}{a_1}K_L - b'\eta \quad \ldots \quad (10)$$

Where
$$b' = \frac{b - .0453\,(1 - E)}{57.3}$$

Values of $-\dfrac{b_1}{a_1}$ and b' are given in Figs. 14 and 15.

Floating Elevators

When calculating tail-setting angles, or dealing with questions of stability, it is often important that the characteristics of the tailplane are known with the elevators floating. This represents the case of flight without control on a machine where the elevator and the control system are statically balanced and without friction. This ideal case could not be realised in practice, but the error involved in making this assumption will usually be small compared with errors due to other causes, unless it is definitely known that the hinge moment is appreciable.

The hinge moment being zero then from 10 (for symmetrical tail)

$$b'\eta = \frac{e}{.25}\cdot\frac{b_1}{a_1} K_L$$

so that

$$\eta = \frac{e}{.25}\,\frac{b_1}{a_1}\,\frac{1}{b'}K_L \quad \ldots \quad (11)$$

and $K_L = a_1 \alpha + a_2 \eta$

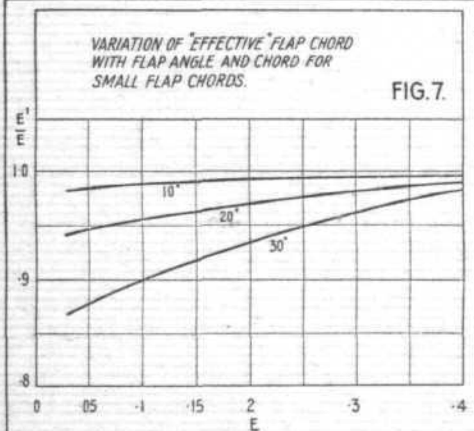

FIG.7.

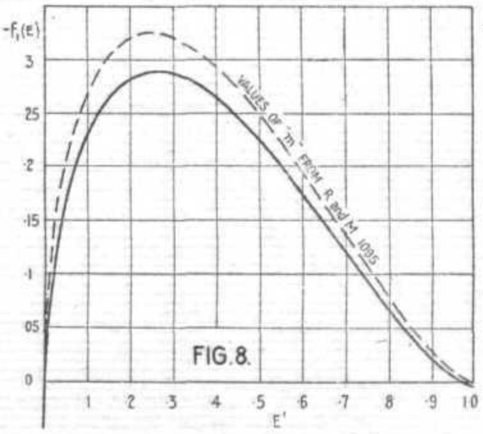

FIG.8.

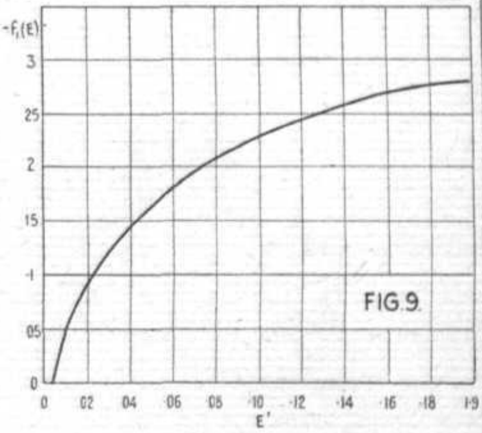

FIG.9.

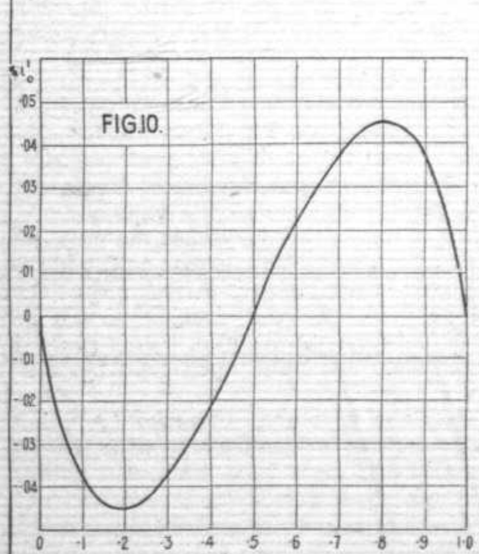

FIG.10.

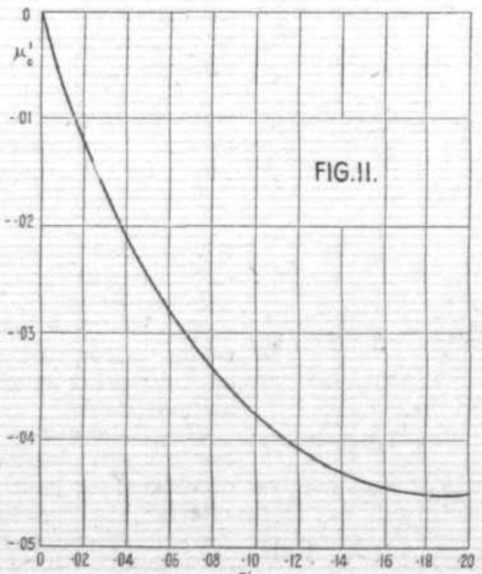

FIG.11.

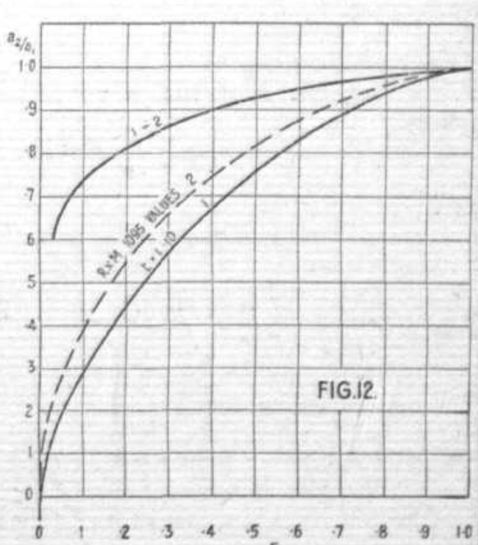

FIG.12.

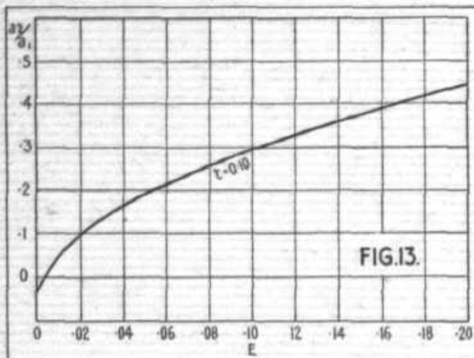

FIG. 13.

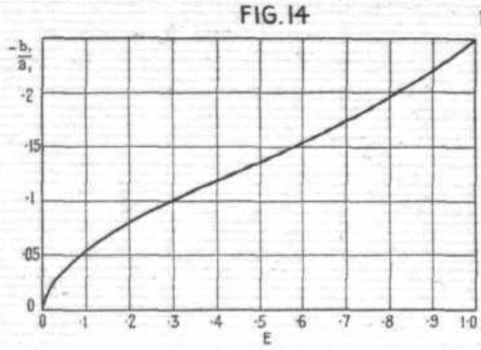

FIG. 14

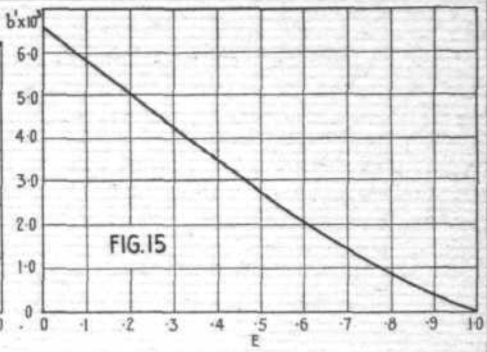

FIG. 15

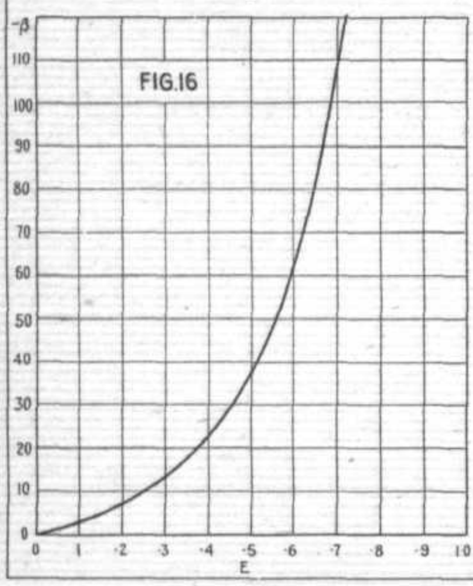

FIG. 16

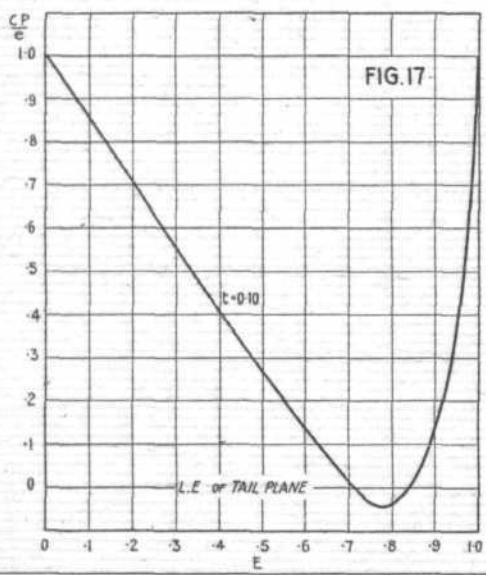

FIG. 17

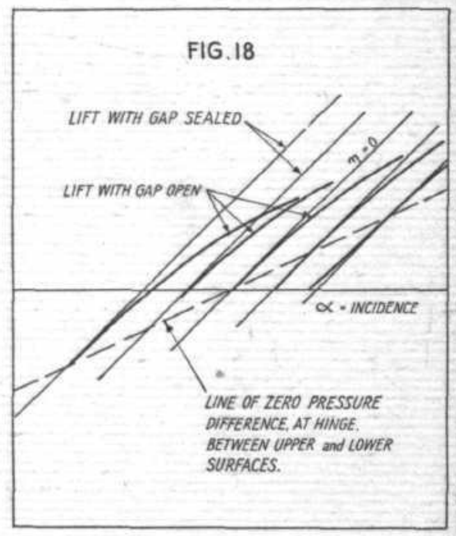

FIG. 18

Substituting for η from 11

$$K_t = a_1 \alpha + \frac{a_2}{a_1} \frac{b_1}{b'} \frac{e}{.25} K_t$$

$$= \frac{a_1 \alpha}{1 - \frac{a_2}{a_1} \frac{b_1}{b'} \cdot \frac{e}{.25}} \quad \cdots \quad \cdots \quad \cdots \quad (12)$$

and $$\frac{K_t}{\alpha} = a_t = \frac{1}{\frac{1}{a_1} - \frac{a_2}{a_1} \times \frac{1}{b'} \times \frac{b_1}{a_1}}$$

Where a_t is the slope of the tail lift curve (elevators floating)

$$\text{or } a_t = \frac{1}{\frac{1}{a_1} - \beta \frac{e}{.25}} \quad \cdots \quad \cdots \quad \cdots \quad \cdots \quad \cdots \quad (13)$$

Where $\beta = \frac{a_2}{a_1} \frac{1}{b'} \frac{b_1}{a_1}$

β is plotted in Fig. 16.

Accepting the approximate relationship for C.P. position as

$$CP = -\frac{Km}{K_t}$$

then

$$CP = e - \frac{Km_0}{K_t}$$

Substituting for Km_0 and K_t from equations 5 and 13, gives

$$CP = e - \frac{\frac{\eta^\circ}{57 \cdot 3} f(t) f_1 (E)}{a_t \alpha}$$

Substituting for η from equation 11 gives :—

$$CP = e - \frac{e}{.25} \frac{b_1}{a_1} \frac{1}{b'} f(t) F_1 (E) \times \frac{1}{57 \cdot 3} \quad \cdots \quad \cdots \quad (14)$$

which is a constant for any value of E. This C.P. position is plotted in Fig. 17 for a 10 per cent. thick tailplane.

Equations 13 and 14 are of great importance in making balance and stability calculations and the constancy of the C.P. with floating elevators makes for simplicity.

It must be borne in mind that all the above relationships apply only to full-span elevators. Elevators, where it is known that the hinge moment is not zero when flying "hands off," require separate and different treatment since equation 10 equals a constant (or some function of incidence, etc.) and not zero as used in this investigation.

This concludes the part dealing with the sealed hinge, and the effect of introducing the gap between elevator and tail plane will now be discussed.

The rigorous treatment of the flow round a wing with a gap and a discontinuity at the gap must of necessity be very involved. An idea of what takes place can be visualised by considering the gap as an orifice through which air can flow when a pressure difference exists between the two sides. This treatment is purely quantitative and no claim is made for the accuracy of any of the assumptions since the matter is obviously too complex for simple treatment. Laws which are established for a continuous aerofoil need not necessarily hold for an aerofoil with a gap.

It has been shown that at a certain incidence, depending upon the flap angle, the lift of an aerofoil vanishes. It follows, therefore, that at some other incidence the pressure difference between the upper and lower surface at the hinge will also be zero. If, now, a small gap is introduced at the hinge, the flow will be zero, and the circulation will be unaffected by the introduction of the gap. This gives a starting point at which the sealed and open gaps agree in characteristics.

If it is assumed that the pressure difference (Δp) between upper and lower surfaces varies linearly with the lift coefficient for both sealed and open gaps, then at constant velocity the flow through the gap will be proportional to the sq. root of the change in lift co-efficient from the reference point where Δp is zero.

The flow through the gap will break down the circulation round the wing by transferring air from the region of high pressure to that of low pressure, tending to equalise the two. Assuming that over a small range, at any rate, the lift is proportional to the incidence, then to get the same lift the incidence of the wing with the open gap must be greater than that with the sealed gap by an amount proportional to the loss of circulation. There is no evidence to show that at the same lift coefficient the pressure difference at the hinge will be the same with open and sealed gaps—it seems doubtful that such would be the case. It becomes, therefore, impossible to obtain a simple expression for the quantitative effect of the gap. Probably within reasonable limits the expression for incidence would take the form of :—

$$a_g = a_s + K_2(K_L - K_{L0})^n \quad \ldots \quad \ldots \quad (15)$$

where a_s = Incidence with sealed gap.

K_L = Lift coefficient.

K_{L0} = Lift coefficient at which the pressure difference between upper and lower surfaces is zero.

K_2 = Constant which involves the air density, size of gap and the coefficient of discharge.

A better picture of what is happening is given by Fig. 18. The position of the line of no pressure difference across the gap suggests that if n (in equation 15) is greater than 1.0 the slope of the lift curves (for positive K_L) will be more nearly the same for negative flap angles and that the slope will become less with negative flap angles. If $n = 1.0$ then the lift curves will all be parallel and the no-lift angle will have a straight line relationship with flap angle.

From a preliminary examination of the V.D. tunnel tests (Ref. 4) on a metal aerofoil fitted with a metal flap it would appear that n is slightly greater than 1.0. It is assumed that the gap between the flap and the forward portion was not sealed on these tests which seems reasonable. Unfortunately no data about the gap have in the past been given in any of the reports examined by the writer so that information on this important question seems to be non-existent. Now that the high pressure tunnel tests are becoming available in increasing numbers and also that flaps are receiving long awaited attention, it is reasonable to expect that tests which show the effect of the gap reproduced to scale in the model will have the necessary data included in the report.

At first sight it might seem that the effect of the gap on the lift of the system is perhaps a refinement hardly worth considering. If, however, the effect is as great, in some circumstances, as the writer anticipates it to be, it has a very important bearing on the question of stability for here the slope of the tail lift curve becomes of paramount importance since it is a direct measure of the restoring force when the aeroplane is disturbed.

In interpreting the above remarks, it must be remembered that the width of the gap and its coefficient of discharge considered as an orifice are a measure of the loss in efficiency, so that where it is impossible or undesirable to seal it, the best alternative is to make the gap as small as practical, and the air path through the gap as long as possible By so doing the resistance to flow through the gap is increased and the lift improved without any additional weight which would adversely affect the stability on account of moving aft the C.G. of the aeroplane.

LIST OF REFERENCES.

1.—The design of aerofoils and the prediction of characteristics. W. R. Andrews. "Flight," 26th Oct. 1933.

2.—R. & M. 910. A theory of thin aerofoils. H. Glauert, M.A.

3.—R. & M. 1095. Theoretical relationships for an aerofoil with hinged flap. H. Glauert, M.A.

4.—N.A.C.A. Report 260. The effect of a flap and ailerons on the N.A.C.A. M.6 aerofoil section. George J. Higgins and Eastman N. Jacobs.

STATIC THRUST AND MODEL EXPERIMENTS

The article by Dr. G. Lachmann on Airscrew-Engine Combinations published on August 29 and September 26, 1935, has brought the following defence of model tests. The authors are engaged in the Aerodynamics Department of the National Physical Laboratory, Teddington.

By C. N. H. LOCK, M.A. and H. BATEMAN, B.Sc.

IN his article on "Airscrew-engine combinations and their effect on the take-off" in *Flight*, August 29th, Dr. Lachmann makes a comparison between the static thrust efficiency deduced from model experiments (R. & M. 829) and that based on American full scale tests. He deduces from Figure 3 of that article that "the model tests become entirely unreliable and misleading." We consider this statement unduly severe on the model tests even if it is applied to the worst case, i.e. to airscrews of high pitch at very low rates of advance. Actually the large difference between the model and full scale curves of Dr. Lachmann's paper represents not primarily a divergence between model and full scale, but almost certainly a result of differences of number of blades and of blade section. In the attached Fig. 1 the curve for the 4-bladed airscrew with R.A.F. 6 sections corresponds to Dr. Lachmann's curve based on R. & M. 829. Unfortunately we have been unable to discover the exact design of screw corresponding to the "American full scale" curve but it almost certainly refers to a 2- (or possibly 3-) bladed screw having blade sections resembling Clark Y, which is now the usual type of section for airscrews. The figure shows three points (triangles) corresponding to some recent tests at N.P.L. of model screws having sections resembling Clark Y, and shows clearly that the discrepancy between model and full scale results in Dr. Lachmann's figure might well result from change of blade section and from 4 blades to 2 blades. These final model results are as yet unpublished and were unknown to Dr. Lachmann at the time his paper was written. Additional evidence is furnished by tests of full-scale metal propellers having R.A.F. 6 and Clark Y sections (N.A.C.A. Report No. 378). These propellers have a blade width 26 per cent. less than that of the N.P.L. models. Unfortunately η_s is very sensitive to change of blade width and number of blades, and for this reason the comparison of these full scale results with the model tests is made on a different basis in Fig. 2. The type of coefficient plotted in Fig. 2 is superior in this respect to η_s and also in the fact that it serves to determine static thrust directly from torque, which will be known approximately for a given engine even if the revolutions or power are unknown. The model results of Fig. 1 are here reproduced together with results of a second model screw of approximately Clark Y section and may be compared with the N.A.C.A. full-scale results. This figure shows clearly that, in spite of the differences of section and blade width of the N.A.C.A. and N.P.L. screws, the difference between model and full-scale results for pitch ratios between 0.9 and 1.15 is much less than that due to variation of blade section.

Finally, it is necessary to comment on Dr. Lachmann's

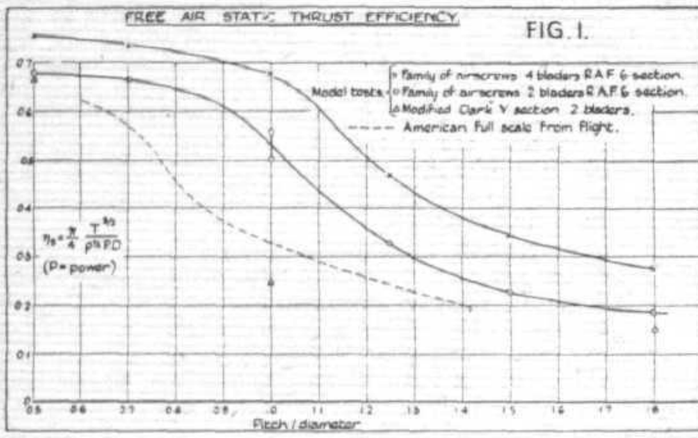

FIG. 1.

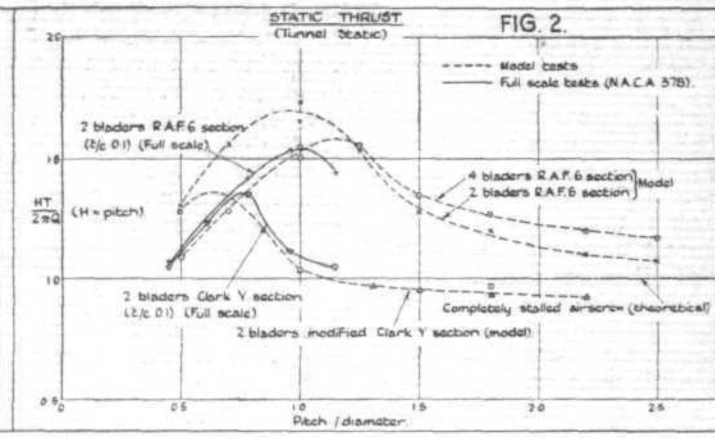

FIG. 2.

criticism of wind tunnel tests that they fail to reproduce true free air static conditions. It must be admitted that the tables of R. & M. 829 are in error to the extent that the performance figures stated to correspond with $J = 0$ actually correspond to tunnel static for which J has values between 0.05 and 0.18. This is made clear in R. & M. 1153 which also provides evidence that the performance of an airscrew at zero forward speed in free air may be derived from wind tunnel observations with fair accuracy by extrapolation (except where the stall takes place at or very near static). The curves of Fig. 1 for the model screws have been obtained in this way, while those of Fig. 2 refer to results at tunnel static.

The final conclusion appears to be that when an airscrew is in the region of the stall, it is essential when estimating its performance, to use data referring to the correct blade section just as much as in the stalled region of an aerofoil. This does not, of course, avoid the necessity for making some allowance for scale effect and tip speed effect, but the above evidence suggests that this is not very important.

TECHNICAL LITERATURE

A Welcome Second Edition

Aeroplane Structures. By A. J. Sutton Pippard and J. Laurence Pritchard, Longmans, Green & Co. 21s. net.

ALL students of the structural side of aeroplane design have long been clamouring for a new edition of this invaluable book. First published in 1919, it embodied all the experience gained by its authors in the war-time equivalent of the Airworthiness Department, of which Mr. (now Professor) Sutton Pippard was the head, and Captain Pritchard was a senior officer. The new edition has been revised with the special object of "helping the student of aeronautical engineering to obtain a logical and thorough knowledge of the basis of aeroplane design." To that revision, Professor Sutton Pippard brings the ripe experience of instructional work gained during his successive tenures of the Chairs of Civil Engineering at Cardiff, Bristol and now at the Imperial College in the University of London. Captain Laurence Pritchard contributes an intimate knowledge of the needs of aeronautical students, whose welfare is well known to be one of his special cares amongst his many duties as Secretary of the Royal Aeronautical Society.

The second edition of "Aeroplane Structures" is a marked improvement on the first, by re-arrangement, by the addition of new material and by the elimination of matter now readily available in other forms. There has been such rapid progress in the art and science of aeroplane construction during the last sixteen years that the authors must have found it difficult to keep the book within reasonable dimensions. To do this they have concentrated on fundamental principles and methods of strength determination, leaving it to the illustrations rather than to the text to show examples of present-day construction.

The calculation of the strength of members under combined lateral and end loads is essential to the design of the spars of a biplane, and the chapter on this subject includes many generalised and modified forms of the Theorem of Three Moments, including the polar diagrams due to H. B. Howard. In the stressing of three-dimensional frames attention is called to the useful tension coefficients due to R. V. Southwell, while a special chapter deals with the application of strain energy methods to the calculation of redundant structures. Much of the recent research and development work on this subject has been carried out by Professor Pippard himself, and he has already written an authoritative book covering the general application of strain energy to structural calculations and design.

Subjects of which little or nothing was known at the time of the first edition are thin metal construction and problems due to flexibility in aircraft structures. The chapters on both these subjects are useful summaries of the present state of theoretical knowledge, with references to the original reports and papers utilised.

A review without criticism is like an egg without salt, and following are a few points which call for adverse comment. Not enough attention has been paid to the definitions of load factors and stresses as used in the recently revised British airworthiness system. The proof factor does not appear to be mentioned at all, nor is it made clear that the calculations for the ultimate factor are directed to reproducing as closely as possible the results which would be attained in a destruction test on a standard structure in which all strengths and thicknesses of materials were the minima permissible. When dealing with fuselage frames it is, of course, usual to assume a pin-jointed structure in determining the loads in the members, but with stiff-jointed forms of construction it is common practice to make some allowance for fixity of the joints in calculating the strength of struts. Mention of stiff-jointed frames leads us to the paragraph on welding, in the chapter on detailed design. However bright the future of spot-welding for joining thin sheets, the most important present use of welding surely lies in tubular construction of fuselages, wing ribs, etc.

This book can be thoroughly recommended to all students of aeroplane structures and to those responsible for teaching them. All of it will come within the scope of those taking an honours course at a University and most of it will be readily understood by much less advanced students. No aircraft technical office can afford to be without a copy for reference purposes. To sum up, this second edition of an already famous book should become the standard textbook on aeroplane structures throughout the British Empire.

H. A. M.

THE AIRCRAFT ENGINEER

SUMMARIES OF AERONAUTICAL RESEARCH COMMITTEE REPORTS

REPORTS published by His Majesty's Stationery Office, London, which may be purchased directly from H.M. Stationery Office at the following addresses: Adastral House, Kingsway, W.C.2; 120, George Street, Edinburgh; York Street, Manchester 1; St. Andrew's Crescent, Cardiff; 15, Donegall Square West, Belfast; or through any ordinary bookseller.

PRESSURE DISTRIBUTION ON WINGS WITH AILERONS. By W. L. Cowley, A.R.C.Sc., D.I.C., and G. A. McMillan, M.Eng., of the Aero-Dynamics Department, N.P.L. R. & M. No. 1625. (73 pages and 20 diagrams). May 14, 1934. Price 3s. 6d. net.

The present tests were made in connection with the design of a large six-engined commercial flying boat. This aircraft was under construction by the Supermarine Aviation Works when the Government decided that it should be abandoned. Tunnel work was also in progress at the time and although most of the investigations could not be completed, several parts of the work should prove useful, in general, in aircraft design. It is intended to issue a report of these sections. The present report deals with the pressure distribution over the wings, and was undertaken for the purpose of investigating the possibility of loss of effectiveness of the aileron due to the twisting of the wings under the aileron loads.

Tests were carried out on a one-twenty fourth scale model of the wings at angles of incidence of $-4°$, $0°$, $4°$, $8°$, $12°$ and $16°$, and the angles of the ailerons in each case were $-5°$, $0°$, and $15°$. The wind speed of test was 75 ft./sec.

From the measured pressures over the wings and ailerons the loads and positions of centre of pressure at each section of the wings and ailerons were calculated under each condition of test.

WIND TUNNEL INVESTIGATION OF THE COOLING OF AN AIR-JACKETED ENGINE. By A. S. Hartshorn, B.Sc. Communicated by the Director of Scientific Research, Air Ministry. R. & M. No. 1641. (55 pages and 18 diagrams.) June, 1934. Price 3s. 6d. net.

Early in 1928 the possibilities of forced draught cooling for an air-jacketed engine were put forward by Mr. R. McKinnon Wood in an unpublished paper* and the present report gives the result of some wind tunnel tests made in connection with this scheme.

Range of Investigation.—Preliminary tests were made on a 6 in. diameter cylinder with and without fins, to find both the rate of heat dissipation to be expected from a jacketed cylinder and the power required to force the air through the jackets.

Other measurements were made on one of the finned cylinders with jackets removed or replaced by rear guide vanes and on a model fuselage of streamline form fitted with internal ducts and gauze to represent the resistance of the jacketed fins. The inlet and outlet passages were modified to produce :—

(a) A flow induced solely by the motion of the body relative to the outside air.

(b) No naturally induced flow, a forced flow being supplied by a separately driven fan.

(c) A composite flow partly induced and partly forced.

Measurements were made of the cooling velocity through the ducts, the drag of the body, the thrust and torque of the fan. The airscrew slipstream was not represented.

The heat dissipation from an air-jacketed cylinder was found to agree reasonably well with the formula given by Osborne Reynolds, using the skin friction coefficient applicable to pipe flow. The power required was about twice that due to skin friction for the finned cylinders and about three times for the plain cylinder.

The tests show that with a cowled system such as a Townend ring the power required for a given rate of heat dissipation can be almost halved by adding suitable rear guide vanes.

For any practical installation the power expended in cooling can be considered as the sum of (1) a necessary loss incurred across the cylinders, and (2) a parasitic loss made up of losses in the entrance and exit ducts and of the effect of the duct on the drag. For a well designed duct this parasitic loss was found to be equal to or slightly less than the necessary loss through the jackets. Additional losses are incurred at top speed if the cooling flow increases with speed.

The data of this report are sufficient to estimate the total excess power required to provide a given rate of heat dissipation for an air-jacketed installation. This power is considerably less than with a normally cowled engine, and can be decreased still further by the incorporation of a method of regulating the cooling flow over the cylinders.

* R. McKinnon Wood. On a proposed forced draught system of cooling aero engines. T.2582. (Unpublished). February 1928.

EXPERIMENTS ON THE WHIRLING ARM. YAWING AND ROLLING MOMENTS ON THE HORNBILL AND VARIOUS AEROFOILS. PRESSURE DISTRIBUTION AND FLOW TESTS ON R.A.F.15. By A. S. Halliday, B.Sc., Ph.D., D.I.C., and C. H. Burge, of the Aerodynamics Department, N.P.L. R. & M. No. 1642. (21 pages and 43 diagrams.) August 11, 1934. Price 1s. 6d. net.

The Hornbill was selected some years ago on account of reports of its good behaviour at low speeds and experiments on a model of it form part of the general investigation of the stability and control of aeroplanes at and beyond the stall. Reports and Memoranda* on full scale and model experiments on the Hornbill have already been published. The data from models in these publications consist of the derivatives due to sideslipping and rolling; the rotary derivatives due to continuous rotation in yaw were then required to complete the study of the lateral stability of the Hornbill. For comparison a number of other models have been tested, mostly of the simple aerofoil type.

The experiments cover a range of incidence from $-8°$ to $40°$. Yawing moments on the Hornbill were measured with rudder central, at $\pm 15°$ and at $\pm 30°$, with tailplane removed, with fin and rudder removed and with the complete model yawed $\pm 2\frac{1}{2}°$ and $\pm 5°$.

The rolling experiments were more comprehensive. Besides experiments on the Hornbill, results are also given for a number of aerofoils, R.A.F.15, R.A.F.31, R.A.F. 15 with swept-forward wing tips, R.A.F.31 with tip slots open and with one open and one closed in turn. Results are also given for R.A.F.15 yawed $\pm 5.05°$.

R.A.F.15 aerofoil has been pressure plotted on the whirling arm over approximately the same range of incidence as for the measurements of moments and for $\pm 5°$ of yaw. The flow over the upper surface of R.A.F.15 has been investigated by means of wool streamers at 30ft./sec. and by smoke at about 7ft./sec. and for comparison the flow determined in a wind tunnel at 30ft./sec. using streamers. With the streamers the flow was also noted when the aerofoil was yawed $\pm 5°$.

When a region of "dead air" is formed above the upper surface of the wings, the lateral acceleration of the aeroplane has an important effect on the aerodynamic rolling moment on the wings. This acceleration depends on the centrifugal component due to rotation in yaw and on rate of change of sideslip; both these factors will therefore influence the rolling moment, and must be taken into account in the study of stability and control beyond the stall. The measurements in this report include the effect of the centrifugal acceleration but do not separate it from the effects of the velocity gradient from tip to tip of the wings. There is therefore no means of estimating the moment when the lateral acceleration includes a part due to rate of change of sideslip. This presents an additional difficulty in the treatment of stability of a stalled aeroplane on classical lines. At the same time the reversal in the sign of the rolling moment due to yawing may account in part for the observed stability of the Hornbill above about 23° incidence, since this reversal checks the divergence which would otherwise be set up by the yawing moment due to rolling of the wings.

* R. & M. 1422. Experiments on the Hawker Hornbill Biplane.—S. B. Gates and other.

AN ANALYTICAL SURVEY OF THE EFFECT OF MASS DISTRIBUTION ON SPINNING EQUILIBRIUM. By S. B. Gates, M.A., and R. H. Francis, M.Sc. Communicated by the Director of Scientific Research, Air Ministry. R. & M. No. 1644. (11 pages and 56 diagrams.) September 26, 1934. Price 1s net.

Emphasis has been laid at various times on the importance of C.G. position and of the inertia differences (A—B) and (C—A) on spinning equilibrium. This work is an attempt to get a rough but comprehensive picture of the whole matter, using such spinning balance data for monoplanes and biplanes as are available.

Charts have been constructed and are arranged in such a way that a designer who knows the layout of his wings and the mass distribution can make a quick estimate of the order of the yawing inertia couple which the combination will give, and also of the effect of changing the mass distribution by weighting either the wings or the fuselage. One set of diagrams show the effect of weighting the wings, and the other that of weighting the fuselage.

The survey confirms the bad inertia wing combinations on which emphasis has previously been laid, but puts these cases in their proper perspective, particularly as regard wings which are not "stable" or "unstable" throughout the whole of the spinning range. As the yawing inertia couple does not usually much exceed 10, which is considerably less than the difference between a good and bad body in flat spins, good body design is a more powerful safeguard than attention to good mass distribution. Slots may increase the pro-spin yawing inertia coupling by over 10 if the fuselage is light.

REPORT ON PUSS MOTH ACCIDENTS. By the Accidents Investigation Sub-Committee. R. & M. No. 1645. (32 pages and 15 diagrams.) March, 1935. Price 1s. 9d.

The Puss Moth is a high-wing cabin monoplane designed and constructed by the de Havilland Aircraft Company, Ltd. It has been flown in all parts of the world in every type of weather, has been used by pilots to make a number of record flights and has flown the Atlantic. This monoplane has, in fact, an excellent record, but there have been some nine structural failures in the air over a period of about three years. Considerable difficulties were experienced in attempting to explain these accidents which occurred all over the world, so that they were referred to the Accidents Investigation Sub-Committee of the Aeronautical Research Committee for a careful investigation.

At an early stage of the investigation it was found impossible to provide a completely convincing explanation of the sequence of events in any of the series of accidents, or to suggest what was the primary cause that had resulted in the structural failure in the air. A series of parallel investigations was accordingly put in hand. None of the earlier enquiries provided an adequate explanation of the accidents, and the possibility of flutter of one of the control organs was considered.

When possible, the wreckage was returned to England and inspected carefully by experts. Amongst the wreckage so collected were eight wings from different aeroplanes which, when the breakages were analysed by the Inspector of Accidents, showed a number of common features. These suggested that the wings must have broken in flutter, because in no other way could an explanation be found of the large forces required to break the wing structure in this manner. Wind tunnel experiments on a properly proportioned model showed that a movement in flutter of the type required to give this kind of breakage was present, and this was considered by the Accidents Investigation Sub-Committee to support the most important evidence of the actual breakages found in the wreckage. The Sub-Committee have accordingly reported that the failure of the wings in several of the accidents, and possibly in all, has been caused by violent racking of the wings, and have pointed out that the particular design of the bracing of the wings by Vee struts will, if wing flutter occurs, give rise to a yawing component tending to rack the wings.

During the course of their investigations, it seemed likely that tail flutter might have caused the damage in some of the accidents but not in others, because in the latter cases the tail part of the structure was found practically undamaged after the accident. Special wind tunnel experiments were made on the tailplane and elevators, and it was found that flutter of both the rudder and elevators was possible under certain conditions. This work was undertaken at an early stage and as a consequence the rudders of Puss Moths are now all mass balanced. However, after this modification had been introduced two further accidents occurred on machines which were known to have balanced rudders.

The Sub-Committee have put forward the following recommendation to the Air Ministry :—

> An increase in the flying speeds of aeroplanes has introduced problems of design distinct from those of strength. The aircraft structure is necessarily flexible and relative movement of its component units takes place under aerodynamic loading. If such movements are excessive, flutter and other dangerous phenomena are liable to occur at speeds within or only a little beyond the normal speed range.
>
> The Sub-Committee are of opinion that the design of modern high speed aircraft involves problems of flexibility as well as strength and the time has arrived when routine calculations or experiments should be made for each design to cover the possibilites of failure due to the interaction of structural distortion and aerodynamic loadings.

THE HYDRODYNAMIC FORCES AND MOMENTS ON SIMPLE PLANING SURFACES, AND AN ANALYSIS OF THE HYDRODYNAMIC FORCES AND MOMENTS ON A FLYING BOAT HULL. By W. G. A. Perring, R.N.C. and L. Johnston, B.Sc. Communicated by the Director of Scientific Research, Air Ministry. R. & M. No. 1646. (23 pages and 31 diagrams.) February, 1935. Price 1s. 6d. net.

An analysis has been made of the force and moment measurements for a plate planing on the water and expressions have been written down showing how these water reactions vary with the aspect ratio of the plate and also with the angle of

vee of the planing bottom. *The effect of longitudinal and transverse curvature* of the planing surface on the forces and moments has also been investigated. In addition the forces and moments have been related to the area of the projected still water surface.

Analysis shows that the forces and moments on a planing surface can be represented by relationship similar to those of an aerofoil. The lift was found to vary directly with incidence, the slope of the lift coefficient curve being a function of the aspect ratio of the form $C\,A^{0.42}$. A similar relationship was also found to hold for vee shaped surfaces, the effect of the vee angle being to modify the coefficient C. The analysis of the drag showed that the forces on a planing surface could be regarded as comprising a force normal to the plane and a frictional force along the plane, which varied with the Reynolds number of the test. *For flat rectangular surfaces* the centre of pressure was approximately at 0.73 of the immersed length from the trailing edge, and expressions were also found showing how the position of the centre of pressure varied with longitudinal curvature, and with $(1 - \lambda)$ the ratio of the immersed lengths of chine and keel of a vee-shaped surface. The analysis was extended to obtain relationships which enable the lift, drag, and pitching moments of flat or vee-shaped planing surfaces to be calculated from expressions based on the projected still water planing area.

THE INFLUENCE OF PICKLING ON THE FATIGUE STRENGTH OF DURALUMIN.† By H. Sutton, M.Sc., and W. J. Taylor. Communicated by the Director of Scientific Research, Air Ministry. R. & M. No. 1647. (4 pages.) February 8, 1935. Price 3d. net.

All the pickling treatments investigated effect a reduction of the Wöhler fatigue limit of duralumin, but the reduction due to treatment C followed by immersion in boiling water is very small and within the range of scatter of the fatigue test results. Immersion in boiling water appears to remove to a substantial degree the injury in the surface layer resulting from pickling. Pickling treatment C reveals the macro-structure of duralumin parts and appears suitable for the examination of pieces at various stages of manufacture for defects such as fatigue cracks, laps and discontinuities. *It is also of service for examination of the structure of ingots*, observation of flow in forgings and stampings, and for the inspection of used parts for fatigue cracks.

* Pickling Treatment C.
(1) Immerse in hot water (boiling water used here).
(2) Transfer directly to bath containing 4 parts of 10 per cent. sulphuric acid (vol.), 1 part hydrofluoric acid, at normal temperature,* immerse for 3 minutes with constant movement, rinse in cold water.
(3) Immerse in cold 50 per cent. nitric acid for 1 minute, rinse in cold water, wash in hot water and dry.

† Abstract only of paper published in *Journal of the Institute of Metals*, 1934 Vol. LV, No. 2, pages 149–158.

* Developed by Messrs. High Duty Alloys, Ltd., Slough, and communicated to authors.

† 20 deg. in the authors' experiments.

COMPARATIVE MEASUREMENTS OF TURBULENCE BY THREE METHODS. By the Staff of the Aerodynamics Department, National Physical Laboratory. R. & M. No. 1651. (17 pages and 8 diagrams.) October 25, 1934. Price 1s. 3d. net.

There are three methods of measuring turbulence in use at the National Physical Laboratory, namely, (a) the hot wire method, (b) the ultramicroscope method, and (c) the spark method, all of which are fully described elsewhere. Of these the first is the oldest and has been most extensively used; the other two are comparatively new. The essential distinction between the methods is that the hot wire record fluctuations at a point fixed in space, whilst the other two methods record the motions of elements of fluid moving with the stream. The latter are, therefore, less influenced by the frequency of the velocity fluctuations at the point of observation. In view of these differences it was important that comparative measurements should be made by the three methods under identical flow conditions, in order to establish the accuracy *obtainable in each case. Accordingly experiments were made in an* airstream in which different degrees of turbulence were obtained by passing the stream through a contraction.

Air from a centrifugal fan was led through a long pipe to a large chamber communicating with the entrance to the contraction. The cross-section of this chamber was made large to obtain a low velocity of approach, and contained a honeycomb which eliminated any large swirl. The flow was further stabilised by a bell mouth leading to the inlet of the contraction. Large disturbances in the flow were generated by a grid of square mesh placed across the stream in the parallel portion of the inlet. Immediately behind the grid the vortex system so formed was regular, and by the time the stream reached the first position for measurement the vortices had become fairly well diffused.

Observations of the longitudinal component of turbulence by the three methods and of the lateral component by the spark method were made at points on the axis at positions in the inlet and outlet of the contraction.

The ratio of Root Mean Square values to maximum values obtained by the hot wire and spark methods are in good agreement, as also are the ratios of the turbulence in the inlet and outlet by all three methods. Any of the three methods, therefore, gives reliable indications of the comparative turbulence in different flows. The type of experiment chosen in the present case was of a particularly drastic nature for in the inlet the flow was very disturbed whilst in the outlet the disturbances were very small. In consequence the accuracy is not high, but it is probable that the data are accurate to within $\pm$ 10 per cent., *since they are in most cases the* means of several measurements. The accuracy of the R.M.S. values is probably higher than this. The absolute values obtained by the three methods are in less satisfactory agreement, the ultramicroscope and spark methods giving higher values than the hot wire.

EXPERIMENTS ON SERVO-RUDDER FLUTTER. By W. J. Duncan,* D.Sc., A.M.I.Mech.E., D. L. Ellis, B.Sc., A.R.T.C., and A. G. Gadd, of the Aerodynamics Department, N.P.L. R. & M. No. 1652. (30 pages and 9 diagrams.) September 11, 1934. Price 2s. net.

There have been several occurrences of flutter of servo-controlled rudders on full-scale, and it was decided to investigate this type of flutter on a model provided with a flexible fuselage. The investigation aimed at finding methods for preventing the flutter. The experiments were all carried out on one model, but a large number of factors affecting the flutter were varied. The principal were the elastic stiffnesses and the inertia coefficients of the system.

In the majority of cases flutter can be prevented by suitable mass loading of the main rudder and servo-flap. There are, however, certain unfavourable combinations of the elastic stiffnesses for which flutter prevention is difficult. It is intended to investigate the case where the flap is geared, so as to act as an aerodynamical balance for the main rudder, which is directly operated.

* Now Wakefield Lecturer on Aeronautics at University College, Hull.

STATIC STABILITY TESTS OF SIX FULL SCALE TWIN FLOAT SEAPLANES. By R. K. Cushing, A. S. Crouch, D.I.C., A.C.G.I., and R. W. Angell. Communicated by the Director of Scientific Research, Air Ministry. R. & M. No. 1653. (20 pages and 13 diagrams.) August 23, 1934. Price 1s. 6d. net.

An important factor in twin float seaplane design is adequate stability on the water. *In a new design this is secured by following the proportions of a previous* successful seaplane, by calculations from design data, or by tests of models in a tank. The requirements which have to be met by service seaplanes are based on extensive service experience.

Experiments have been made to determine the longitudinal and lateral metacentric heights and the capsizing moments on the Vildebeest, IIIF, Osprey, Nimrod, Tutor and Atlas twin float seaplanes. Some of these seaplanes have been tested at both limits of the centre of gravity, the stability of each seaplane has been calculated and the calculations compared with the experimental results.

The experimental results show that the lateral and longitudinal metacentric heights can be calculated with a good degree of accuracy, if the water line is known accurately.

An error of 5 per cent. in the displacement results in errors of 1 per cent. and 2½ per cent. in the lateral and longitudinal metacentric heights respectively, while an error of ½ deg. in attitude results in errors of approximately 3 per cent. and 9 per cent. in the lateral and longitudinal metacentric heights respectively.

The moment required to capsize the seaplanes when trimmed aft is somewhat smaller than the requirements of Air Ministry Specification No. 22 Miscellaneous. It would appear, however, from the general opinion of experienced pilots, that in practice these seaplanes have adequate stability.

SPINNING EXPERIMENTS ON A MODEL OF THE BRISTOL FIGHTER AEROPLANE, INCLUDING THE EFFECT OF WING TIP SLOTS AND INTERCEPTORS. By H. B. Irving, B.Sc., A. S. Batson, B.Sc., and J. H. Warsap, of the Aerodynamics Department, N.P.L. R. & M. No. 1654. (19 pages and 14 diagrams.) February 28, 1935. Price 2s. net.

This report describes the first experiments made at the National Physical Laboratory, *with the new spinning balance*[1] *in which the actual motion of a steady* spin can be represented; and moments about two axes measured, one axis being the axis of rotation, and the other an axis fixed in the body of the model, which may by choice be either the yawing axis or the pitching axis.

The Bristol Fighter aeroplane was chosen for the experiments, because of the very extensive data on its aerodynamic qualities which were available, including quantitative data from full-scale spinning tests.[2]

The range of investigation included the following experiments:—

Centre of Gravity on Axis of Rotation.—Model without slots: effects of rudder, elevator and aileron settings (upper wing only) at 0 deg. sideslip; effect of 5 deg. sideslip; effects of wing tip slots and interceptors on upper wing only (0 deg. sideslip).

Centre of Gravity off Axis.—Comparison with full-scale spins.

The effect of slots on spinning moments make it seem probable that all biplanes with wing tip slots, even though they have considerable stagger (which tends towards stability in roll) will become unstable in roll when the incidence exceeds about 45 deg. Adding weights to the wings of a slotted biplane will accordingly be generally adverse in effect in flat or flattish spins even when there is stagger.

The yawing moments produced by wing tip slots are such that the slots will general tend to accentuate any flat spinning tendency a machine may have, or bring out a latent possibility of the flat spin occurring.

Owing to scale effect the vertical spinning tunnel probably errs in the direction of giving the model a greater margin of safety in spinning than the corresponding *full-scale machine.*

[1] A continuous rotation balance for the measurements of yawing and rolling moments in a completely represented spin.—P. H. Allwork. R. & M. 1579.

[2] Further experiments on the spinning of a Bristol Fighter Aeroplane.—A. V. Stephens. R. & M. 1515. July, 1932.

THE EFFECT OF WING SETTING ON THE WATER PERFORMANCE OF SEAPLANES. By W. G. A. Perring, R.N.C. Communicated by the Director of Scientific Research, Air Ministry. R. & M. No. 1656. (8 pages and 3 diagrams.) August 15, 1934. Price 9d. net.

The forward motion of a seaplane during the take off is opposed by the air and water forces and a change in the wing setting relative to the hull will modify both these forces. Thus if the wing setting relative to the hull or floats is increased, the wing attitude and consequently the wing drag is also increased. At the same time however the change in the air lift will have brought about a decrease in the load on water and therefore a decrease in the water drag. The effect of a change of wing setting relative to the hull on the water performance therefore, is dependent upon the effect that such a change has upon the combined air and water drags. Expressions have been written down for the combined air and water drag of a seaplane and these have been used to determine the optimum angle of the wings relative to the hull so far as the water performance is concerned. The theoretical work has also been supported by calculations showing the effect of wing setting on the water performance of a typical flying boat, these calculations being based upon tank and wind tunnel data.

The optimum wing setting of a seaplane is shown to be dependent upon the running angle of the hull and the aspect ratio of the wing structure, and the best angle in the case of monoplane is greater than the corresponding angle for a biplane of the same aspect ratio. The experimental results are shown to be in very good agreement with the theory.

THE WATER PERFORMANCE OF SEAPLANES. APPLICATION OF TANK DATA TO DETERMINE THE EFFECT OF WIND, VARIATION OF LOADING OR A CHANGE OF AIR STRUCTURE UPON THE PERFORMANCE. By W. G. A. Perring, R.N.C. Communicated by the Director of Scientific Research, Air Ministry. R. & M. No. 1657. (4 pages and 4 diagrams.) October 6, 1934. Price 6d. net.

Tank tests of model seaplanes usually relate to one, or sometimes two load conditions, and the tests correspond to conditions of no wind for some specified air structure. The tank tests, which are carried out at an early stage of the design, are often not directly applicable to the completed seaplane, because of modifications made to the wing structure or to changes in the all-up weight. The effect of wind on the take-off is another problem to which the routine tank test is not directly applicable. Changes in the air structure, or the loading, or the effect of wind, all result in a load on water differing from the load on water during the actual tests, and the effect of any one of the changes can therefore be calculated provided that the effect of variations of load on water can be predicted.

The method outlined in this report is suitable to determine the effect on the take-off performance of a seaplane, of any change in the conditions that involves a change in the load on water.